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PROSPECTS FOR ADVANCED NUCLEAR SYSTEMS

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I. INTRODUCTION

The purpose of this paper is to examine the ultimate performance potential of nuclear fission reactors as power sources for the direct heating and acceleration of a propellant by means of a rocket nozzle. Specifically, we will discuss the features of an "ultimate nuclear rocket" in terms of the unique properties of the fission process, relate these to various vehicle missions, and finally, suggest the prospects for the engineering realization of such systems.

The subject of this paper will not include "conventional" nuclear systems, which for the present purposes are defined as any reactor concept which incorporates nuclear fuel-bearing material in either solid or liquid phase. Thus we will exclude advanced reactors of the solid-fuel heat-exchanger type, molten fuel reactors, and dust fueled reactors. All these systems are in a real sense temperature-limited, either by the melting or varporization point of the solid, or by the boiling point of the liquid.

The reactor systems to be considered are postulated to include some region of the nuclear fuel-bearing medium which in a complete sense is <u>nontemperature limited</u>. In practical terms, this means that some portion of the nuclear fuel is in gas phase; thus, as in chemical engines, it becomes possible to maintain this portion of the fuel medium at temperatures significantly greater than that tolerable in either a liquid or solid state. This implies, of course, that the solid retaining boundaries of this gaseous medium are cooled by some suitable process, again as in conventional rocket engines.

The approach outlined here is sufficiently general to encompass the various gaseous fuel reactor concepts suggested to date. These include the vortex containment technique (see Ref. 1) which requires intimate mixing of the fissionable species and the propellant and the plasma core (see Ref. 2) and coaxial flow (see Ref. 3) reactor concepts, both of which postulate the physical separation of the fuel and propellant. In the vortex technique the heat transfer is achieved by the direct slowing down of the nuclear fission fragments in the propellant gas, whereas in the last two methods mentioned, the fission fragments slow down in the fuel region which then transfers its energy to the propellant principally by thermal radiation. The specific ideas and results to be discussed, however, apply directly to methods which incorporate intimate mixing in gas phase of the fuel and propellant. Although, as already suggested, the other two methods may also be treated by the present approach, these require a reformulation of the problem in terms of the specific physical features of the system, in particular the radiative characteristics of the fissioning plasma medium. In this sense, the present analysis is not complete, but the general results are in gross terms applicable to the radiative heat transfer systems as well. Certainly, the incentives and overall conclusions to be drawn from the present study are generally valid to all such gas-phase fission reactors.

Finally, it should be mentioned, that for purely analytical reasons, the reactor and engine complex utilized in this work are highly idealized models. It is to be expected, therefore, that the numerical results will be optimistic, and in some cases unrealistic. These results will define, however, the upper bounds for any practical system and thereby serve as a measure of excellence for future development.

II. GENERALIZED NUCLEAR ROCKET ENGINE

In order to provide the most general description of a nuclear rocket engine we combine the nontemperature-limited reactor concept with an engine complex that includes a heat rejection device, here conveniently taken to be a space radiator. Figure 1 shows a schematic of the proposed engine. The configuration shown is but one of many possible arrangements. The choice of a particular scheme is not essential to the overall argument; however, the choice of components is, and these will determine some of the general engine characteristics. Consider the reactor. The figure shows the temperature-limited fuel zone in a central location (here represented by a radial array of solid fuel plates) and the nontemperature-limited zone in a surrounding annular region. Beyond the latter, and at each end, is a neutron reflector. The nontemperature-limited fuel bearing regions are represented by a matrix of cylindrical cavities containing gaseous fuel imbedded in a moderator matrix. As already noted, this definition of a gaseous system is most appropriate to the vortex containment method; it may also be applicable, however, to the plasma core or coaxial flow reactor concepts.

The radiator and closed-loop coolant circuit disposes of unsuitable or unavailable forms of energy released in the engine complex such as fission power from the solid (temperature-limited) fuel bearing medium, nuclear radiation, and thermal radiation. Some of this power may be removed also by regenerative cooling by the propellant. In the scheme shown, the propellant is heated from the storage condition (at say enthalpy per unit mass $h_o = 0$) to the value h_s corresponding to the maximum allowable temperature T_s in the solid regions of the reactor; it is then passed into the cavities where it is further heated, by

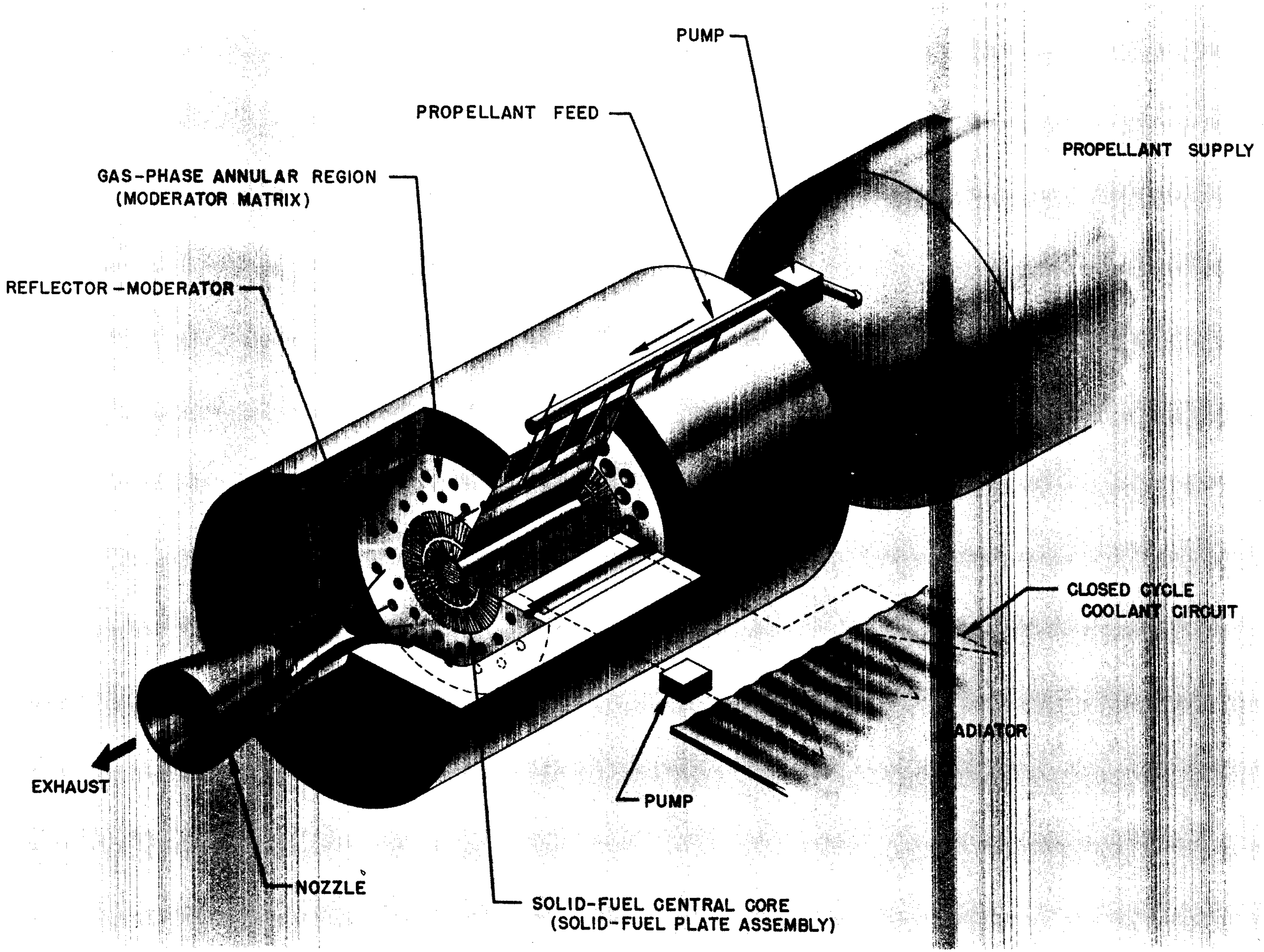

Fig. 1. Conceptual Arrangement of Nuclear Rocket Engine with Nontemperature-limited Fuel-bearing Region and Radiator System

intimate mixing with the gaseous fuel, to enthalpy h_c (or temperature T_c) and finally expanded through a rocket nozzle.

The performance potential of this system may be obtained in terms of the specific impulse I_c, corresponding to the enthalpy h_c, from an analysis of the power balance in the engine (see Ref. 4). If it is assumed for simplicity that the propellant at the stagnation enthalpy h_c is completely expanded in a de Laval nozzle, then $I_c = \sqrt{2h_c}/g$. The power balance for the generalized system which includes a radiator then yields

$$\left(\frac{I_c}{I_s}\right)^2 = \frac{1 + \gamma(1 - f)(1 - \zeta) - \beta\left[\left(\frac{I_c}{I_s}\right)^8 - 1\right]}{f + \zeta(1 - f)} \tag{1}$$

where I_s corresponds to h_s, ζ denotes the fraction of the fission energy which appears as nuclear radiation (and penetrates the gas in the cavities and is attenuated only by the surrounding solids), f is the fraction of the total fission power produced in the solid fuel regions, and

$$\beta \equiv \frac{\sigma\epsilon_c A_c T_s^4}{\dot{m}h_s} \qquad \gamma \equiv \frac{P_r}{\dot{m}h_s} \tag{2}$$

The symbol P_r denotes the power rejected by the radiator, $\dot{m}$ is the mass flow rate of the propellant, σ is the Stefan-Boltzmann constant, and ϵ_c the emissivity, and A_c the surface area of the gas mixture in the cavities.

Figure 2 shows the specific impulse ratio $I \equiv I_c/I_s$ as a function of the thermal radiation parameter β for various values of the solid fission fraction f. These results are for system without radiator ($\gamma = 0$), and as will be discussed shortly, applicable

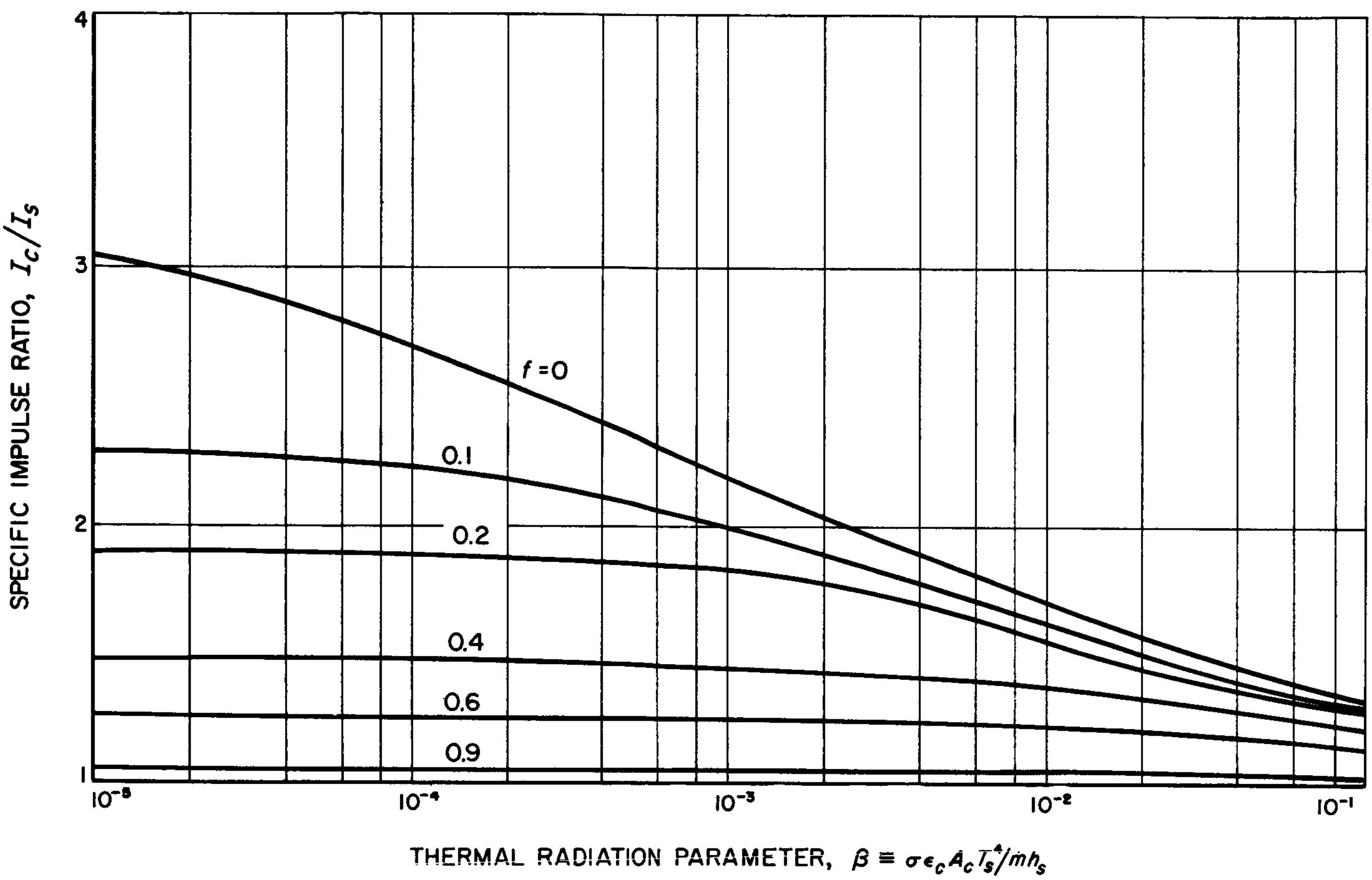

Fig. 2. Specific Impulse Ratio as Function of Thermal Radiation Parameter for System Without Radiator

to the high-acceleration (or ground takeoff missions). Two trends are of significance: first there is a maximum value of the specific impulse ratio for each value of the solid fission fraction f, the largest of which $I = 3.13$ occurs for $f = 0$; second, increasing values of β compromise the rocket performance. The upper bound on I arises because of the quantity ζ (here taken as 0.1). The explanation is given in terms of the power balance. Since the present systems are regeneratively cooled, the enthalpy h_s gained by the propellant in cooling the solid regions must be exactly the fraction ζ of the total enthalpy h_c it gains in passing through the reactor. Thus, $h_c/h_s = I^2 = 1/\zeta$. In the case of hydrogen propellant and a maximum reactor solids temperature $T_s = 2000°K$, this limit yields $I_c = 2210$ sec. When $f > 0$, the effect on the system is to increase the effective value of ζ; the same effect arises also from the thermal radiation produced by the gases in the cavities. Thus we find that even with the aid of a nontemperature-limited reactor concept, the fission process does not yield unlimited performance potential in systems entirely regeneratively cooled.

When a radiator is added, however, the upper bound on I_c/I_s is removed, and arbitrarily larger values can be obtained by simply rejecting a greater fraction of the heat attenuated by the solid. Figure 3 shows the specific impulse ratio for the case $f = 0.6$ (which is appropriate for the low acceleration mission) as a function of the radiator power fraction γ. These results indicate, for example, that if the radiator removes all but one part in a thousand of the heat deposited in the solid, then specific impulse ratios as high as 10 are possible.

From a practical point of view, however, such large values are not attractive. The reason is that increasing radiator capacity brings with it an increasing penalty in radiator weight. Consequently, the higher specific impulse systems are

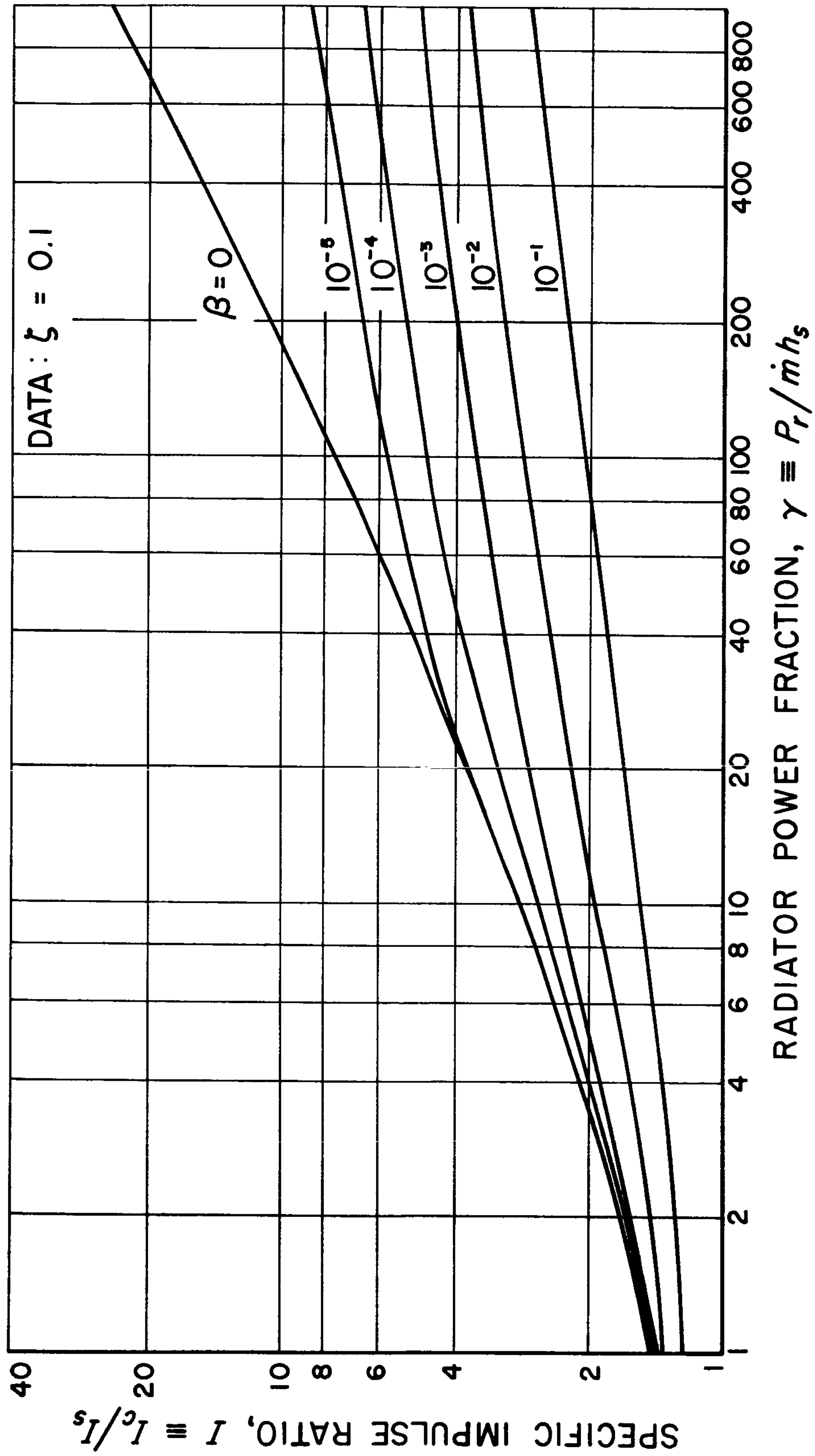

Fig. 3. Specific Impulse Ratio as Function of Radiator Power Fraction for f = 0.6

characterized by engine thrust-to-weight ratios substantially less than unity; thus, application is limited to low-acceleration (interplanetary) missions. The general expression for the engine thrust-to-weight ratio a_e may be written

$$a_e = \frac{I}{\gamma\eta + \frac{1}{\theta}(1+\gamma)} \tag{3}$$

where

$$\theta \equiv \frac{2p_s}{w_s c_s} \quad \text{and} \quad \eta \equiv \frac{\delta c_s}{2\sigma\epsilon_r T_r{}^4} \tag{4}$$

and p_s is the power density in the solid regions of the reactor, w_s, the weight per unit volume of reactor material, $c_s = I_s g$, δ is the radiator weight per unit area, and $(\ )_r$ refers to the radiator properties. The first term in the denominator of Eq. (3) is due to the radiator, and the second is due to the reactor. In the case of the system with radiator, a more widely used parameter is the specific weight α_*, here defined as the ratio of reactor plus radiator weight to the power imparted to the exhaust. Figure 4 shows α_* as a function of the radiator power fraction. The values noted on the figure correspond to $I_s = 700$ sec, $p_s = 30$ w/cm^3, $w_s = 3$ gm/cm^3, $\delta = 5$ lb/f^2, $\epsilon_r = 0.9$ and $T_r = 1000°$K. It is seen that for values of $\gamma < 200$ the specific weights are entirely comparable to nuclear electric systems. For $\gamma < 60$, they are far superior.

When the radiator is omitted from the engine the resulting system, although limited in maximum obtainable specific impulse, is capable of rather large thrusts. For these systems the engine thrust-to-weight ratio is given by: $a_e = \theta I$. It should be noted that the parameter θ may be interpreted as the thrust-to-weight ratio of an all temperature-limited (i. e., solid fuel) reactor operating at maximum temperature T_s. With the constants given above and a power density of 2 kw/cm^3, $\theta = 20$. Thus at the maximum value of the specific impulse ratio, $a_e \simeq 60$.

III. NUCLEAR CHARACTERISTICS

The nuclear characteristics of the reactor may be determined directly from the selected engine parameters, such as f, β, γ, and θ, once the mission has been selected. In general, the specification of the mission is stated in terms of a velocity increment and a payload to be delivered. A detailed trajectory analysis then establishes the propellant and payload fractions, and from this information, one can determine the reactor thrust, power, and size.

The reactor size and power in turn determine the average nuclear fuel concentration, critical mass, and average thermal neutron flux. For the purposes of this analysis a simplified nuclear model was considered adequate. Calculations were performed for a bare, homogeneous reactor using the Fermi age theory (see Ref. 5). A uniform correction was introduced, however, to account for the presence of the cavities which contain the gaseous fuel. Detailed nuclear computations require a selection of the void fraction x and information on the distribution of the fissionable material between the gas and solid phases. This latter is implied in the solid fission fraction f.

Some typical results for the nuclear characteristics are given in connection with vehicle performance data.

IV. APPLICATION TO HIGH-ACCELERATION SYSTEMS

In the case of the high-acceleration missions, performance calculations were made for simple "burnout velocities" of single-staged vehicles traversing drag-free vertical trajectories in a constant gravitational field. Given the velocity increment to be attained and the specific impulse available, the payload fraction π_{pl} is determined. For the present systems, this may be expressed as

$$\pi_{pl} \equiv \frac{W_{pl}}{W_o} = 1 - \lambda(1 + s) - \frac{a_o}{a_e} \tag{5}$$

where λ is the propellant fraction, s the ratio of tank weight to propellant weight, and a_o is the initial vehicle acceleration. Vehicle characteristics based on this relation and the vacuum trajectory equation have been computed for two high-performance missions: first, a booster to place a 300,000-lb gross payload in 300-mi Earth-satellite orbit; and second, a vehicle to launch a manned capsule of the Apollo type from the Earth, land on the Moon, take off from the Moon, and finally return to the Earth.

The first mission was selected to demonstrate the capabilities of advanced nuclear systems as satellite freighters. For comparative purposes it should be noted that this mission is being considered for the Nova vehicle, a 3-stage chemical system of about 5 million lb takeoff weight. Table 1 summarizes some of the principal characteristics of the nuclear vehicle and engine. The engine parameters are listed at the bottom of the table, of particular interest being the cavity void fraction $x = 0.6$, the power density $p_s = 2\ \mathrm{kw/cm^3}$, the "Rover-equivalent" thrust-to-weight

Table 1. Vehicle and Engine Characteristics for Placing 300,000-lb Payload in 300-Mile Earth-Satellite Orbit

System	I_c	T_C	W_o	π_{pl}	W_n	P
Gaseous fuel fraction	Specific impulse, sec	Cavity temperature, °K	Vehicle gross weight, lb	Payload fraction	Reactor weight, lb	Reactor power, Mw
0.94	1160	3800	1,380,000	0.217	54,000	45,000
0.60	967	3100	1,870,000	0.160	88,000	51,000
Chemical	420	—	5,000,000	0.06	Nova	

System	ℓ	N_{FC}	p	ϕ_c	M_F
Gaseous fuel fraction	Reactor diameter, ft	Fuel concentration in gas phase, nuclei/cm^3	Average cavity pressure, atm	Thermal flux in cavity, neut/cm^2 sec	Critical mass, kg
0.94	9.7	11.3×10^{17}	57	1.23×10^{17}	5.9
0.60	11.4	4.95×10^{17}	21	1.95×10^{17}	6.6

Data: $\beta = 10^{-2}$, $s = 0.05$, $\zeta = 0.1$, $x = 0.6$, $a_o = 1.3$, $\theta = 20$, $\phi_c/\phi_S = 1$,

$p_s = 2$ kw/cm^3, BeO Moderator, $\sigma_a^{(F)} = 1000$ barns ratio θ = 20, and the thermal radiation parameter $\beta = 10^{-2}$. This last choice is an estimate for cavity tubes of about an inch in diameter and propellant emissivities of about 0.1. Two systems are reported, one in which nearly all the fissionable material is carried in gas phase, and another in which 60% is in gas phase. An immediate observation to be made is that the system with less fuel in the gas phase exhibits a smaller payload fraction and larger engine characteristics. The gain over the Nova, however, is apparent, in either case. The indicated reactor weights, total power and diameter are not unreasonable for so large a vehicle, but the critical masses are low. This results from the fact that the thermal absorption cross section of the non-fuel components in the reactor was taken to be that of the moderator BeO alone (which is in the order of millibarns). In a practical situation, many foreign materials in the form of structural components and the like would be present, and this would lead to an increase in the critical mass by perhaps as much as a factor of 5. The average thermal neutron flux, on the other hand, is seen to be about two orders of magnitude larger than encountered in current technology. Values of 10^{17} neut/cm^2 sec, however, are generally characteristic of high-performance nuclear propulsion devices, including solid-fuel heat-exchanger types.

The choice of the lunar-landing-and-return mission was selected as a very difficult feat which would exhibit in more conspicuous fashion the incentives for pursuing the developing of these advanced systems. The performance requirements were based on a 33,000 f/s escape velocity from the Earth, followed by a landing and takeoff velocity increment of 9000 f/s each, and a corridor-reentry reserve of about 2000 f/s. It was assumed that all the dead weight in the takeoff vehicle was retained throughout the mission; this includes, reactor, tanks, and payload. The 3-man Apollo capsule weight for a 2-week mission was taken to be 15,000 lb. Table 2 gives the required system characteristics for this mission. The engine parameters used in this calculation are the same as those for the satellite freighter. Some idea of the gains to be realized by utilizing these systems is obtained from a comparison with major chemical boosters under development. It would take roughly 4 Saturn C-2 (3 stage) vehicles in a rendezvous operation to accomplish this mission, or a single 5-million-pound Nova. In either case the payload fraction is about 0.003.

Table 2. Vehicle and Engine Characteristics for Lunar Landing and Return of Apollo Capsule (W_{pl} = 15,000 lb)

System	I_c	T_C	W_o	π_{pl}	W_n	P
Gaseous fuel fraction	Specific impulse, sec	Cavity temperature, °K	Vehicle gross weight, lb	Payload fraction	Reactor weight, lb	Reactor power, Mw
0.94	1160	3800	375,000	0.040	15,000	12,300
0.79	1080	3600	667,000	0.022	28,000	20,000
Chemical	420	—	5,000,000	Nova or 4 Saturns		

System	ℓ	N_{FC}	p	ϕ_c	M_F
Gaseous fuel fraction	Reactor diameter, ft	Fuel concentration in gas phase, nuclei/cm^3	Average cavity pressure, atm	Thermal flux in cavity, neut/cm^2 sec	Critical mass, kg
0.94	6.3	10×10^{18}	500	1.4×10^{16}	14
0.79	7.8	2×10^{18}	93	6.1×10^{16}	6

Data: Δv = 53,000 f/s, $\beta = 10^{-2}$, s = 0.05, ζ = 0.1, x = 0.6, a_o = 1.3,

θ = 20, ϕ_c/ϕ_S = 1, p_s = 2 kw/cm^3, BeO Moderator, $\sigma_a^{(FC)}$ = 1000 barns

V. APPLICATION TO LOW-ACCELERATION SYSTEMS

The incorporation of a radiator into the engine complex leads to considerable versatility in these advanced systems. By adding a radiator they become competitive, at least in principle, with low-thrust nuclear-electric systems. The principal incentive for considering this application is the prospect of performing difficult interplanetary missions at substantial reductions in total trip time, a consequence of the inherent high thrust capability of the gaseous fission reactor.

Some indication of performance potential is obtained from the analysis of two representative cases, a Mars and a Jupiter mission. In each case the mission is to propel a spacecraft along a minimum energy trajectory starting from an Earth satellite orbit at 300 mi, to the target planet where the payload is to be captured in a circular orbit at 1.1 times the planet radius. The requirements for the Mars orbiter are representative of a relatively easy mission, whereas the requirements for the Jupiter orbiter are of a relatively difficult one. The trajectories begin with an escape spiral at the Earth during which period a constant thrust is applied at right angles to the local radius vector from the center of the force field. When the appropriate hyperbolic velocity has been achieved to enter the minimum energy ellipse, the vehicle then coasts, and at the destination planet enters a deceleration spiral until a capture is accomplished.

The vehicle and engine characteristics for the Mars orbiter are given in Table 3 for an initial spacecraft weight of 500,000 lb and various initial accelerations at the Earth satellite orbit. The indicated specific impulses are optimized so as to obtain the maximum payload fraction for the given initial acceleration. The selected

Table 3. Characteristics of 500,000-lb Spacecraft to Deliver a Mars Orbiter

a_o	I_c^*	T_C	t_{total}	W_{pl}	W_n	P	ℓ
units of g	sec	°K	days	lb	lb	Mw	ft
10^{-5}	2450	11,400	2120	203,000	3200	15	3.2
10^{-4}	1890	6200	424	142,000	6200	32	3.9
10^{-3}	1190	4000	274	104,000	5200	32	3.7
10^{-2}	910	2800	263	70,000	22,000	145	5.9
10^{-4}	14,000	-	498	280,000	Nuclear-Electric, 16.6 Kwe, 10 lb/kw		
2.7×10^{-4}	4000	-	332	134,000			
-	420	-	230	40,000	2 <u>Nova</u>		
1.7×10^{-2}	900	-	230	72,000	6 SaR-R (<u>Saturn</u> + <u>Rover</u> + low-thrust <u>Rover</u>)		

a_o	N_{FC}	p	ϕ_c	M_F	a_*
units of g	nuclei/cm^3	atm	neut/cm^2 sec	kg	lb/kw
10^{-5}	-	-	-	-	52
10^{-4}	6.8×10^{18}	540	1.73×10^{14}	2.6	14
10^{-3}	12.8×10^{18}	680	1.12×10^{14}	3.3	2
10^{-2}	1.4×10^{18}	57	11.2×10^{14}	2.0	0.6

*Optimum for maximization of payload fraction.

Data: $\beta = 10^{-2}$, $f = 0.6$, $\zeta = 0.1$, $s = 0.05$, $x = 0.3$, $\theta = 0.3$, $\eta = 16$,

$p_s = 30$ w/cm^3, $\phi_c/\phi_s = 1$, BeO Moderator, $\sigma_a(FC) = 1000$ barns engine parameters are indicated in the footnote, the most important of these being the power density (30 w/cm^3) and the thermal radiation parameter ($\beta = 10^{-2}$). This very low power density was chosen in order that the reactor sizes be large enough to allow a critical assembly. Even then, it may be noted that the reactor for $a_o = 10^{-5}$ will not go critical. The remaining three cases do yield critical systems, but these configurations are not optimum and consequently the fuel concentrations in gas phase are high. Concentrations in the order of 10^{16} nuclei/cm^3 are preferable so that the required cavity gas pressures be reasonable. For these cases the fluxes would be in the order of 10^{16} to 10^{17} neutrons/cm^3 sec. No attempt was made to improve the nuclear characteristics of these systems since the selected reactor model was considered too coarse to warrant refinement. More accurate computational models, coupled with the use of various reflected geometries and flux peaking techniques, and finally the inclusion of fast fission reactions, are expected to yield better nuclear configurations than suggested here.

When compared to scaled-up versions of nuclear-electric systems under development or projected for the near future, it is seen that the payload capability of the present systems, for an initial acceleration of 10^{-4} g is about half that predicted for the nuclear-electric. However, as the initial acceleration increases, the gaseous systems rapidly overtake the electric. Missions at accelerations above about 5×10^{-4} g can only be performed by the gaseous systems. These observations apply in general to missions to the near planets. At these relatively higher accelerations, the trip time is reduced very nearly to the so-called coast period; thus these systems behave very much like spacecraft launched directly from the ground by boosters. With these vehicles the thrust is applied impulsively (in the order of minutes); but this is essentially the characteristic of the gaseous engines for $a_o > 10^{-3}$. As may be seen from the indicated engine specific weights (and Fig. 4), these engines consist primarily of reactor and therefore give high thrust-to-weight ratios and short propulsion times.

If one attempts a more difficult interplanetary mission, as represented by the Jupiter orbiter, the performance of the gaseous system falls off badly. For example, if one were to apply the same engine characteristics to this mission as in the Mars, then the resulting system would be incapable of making the trip. In order to achieve reasonable performance it is necessary that the radiation parameter β be reduced from 10^{-2} to 10^{-4}. Then again gains are made in reducing trip times (Table 4). This is of course most important if the mission is to be a manned flight. It is interesting to note that for the Jupiter mission the optimum specific impulses are considerably higher than in the case of the Mars orbiter. This is due to the fact that by reducing the thermal radiation parameter, the penalty in radiator weight is also reduced, and markedly higher specific impulses can be achieved before the attendant engine weight cuts substantially into the payload capability. Further reductions in β would, of course, result in mission capability into the 10^{-3} to 10^{-2} range of initial accelerations.

These results indicate that in missions to the more distant planets, the gaseous systems are not competitive with the nuclear electric unless values of the radiation parameter much less than 10^{-3} can be achieved. In that event, these systems not only compete favorably, but offer considerable savings in trip time.

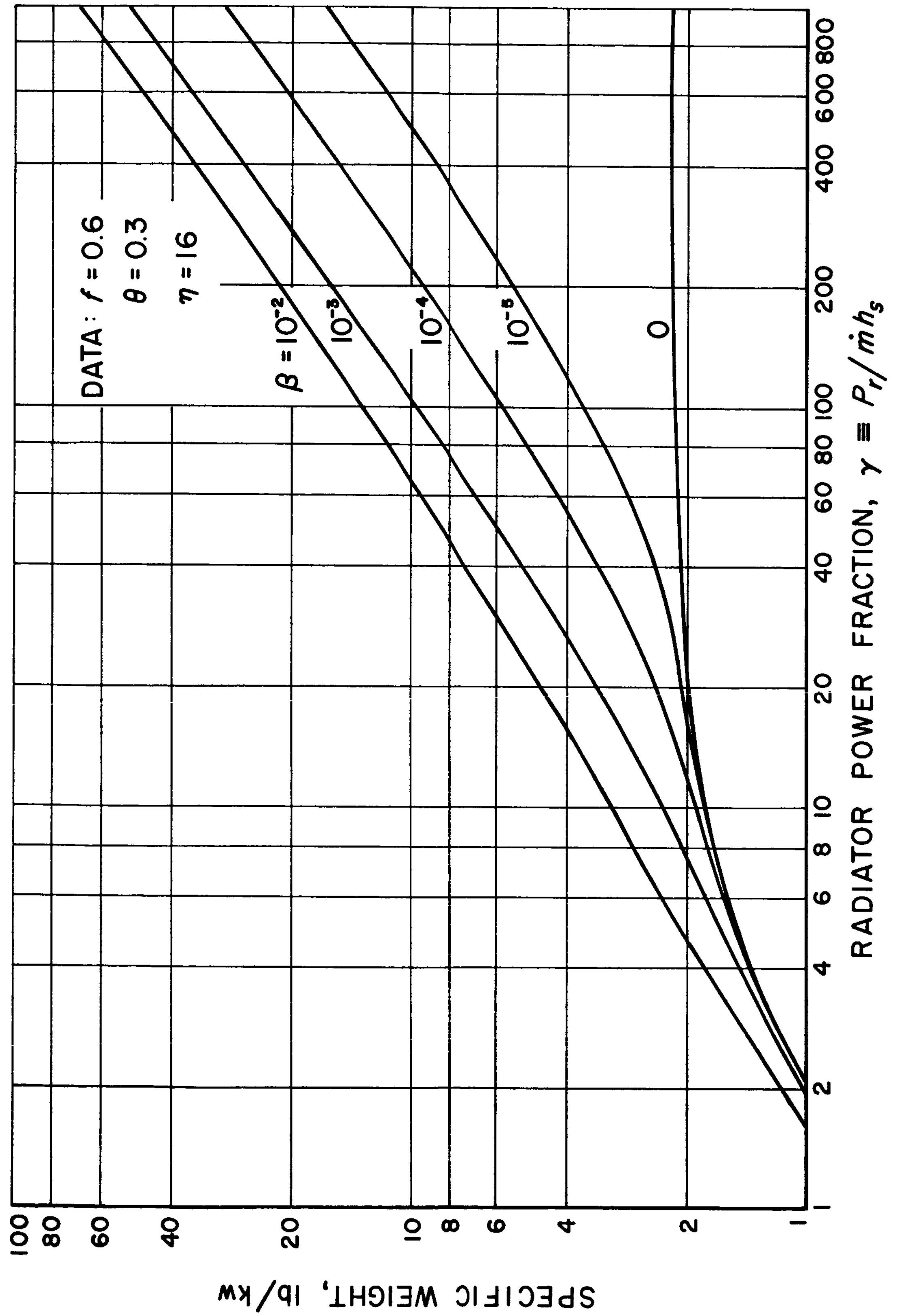

Fig. 4. Engine (reactor plus radiator) specific weight as function of radiator power fraction

Table 4. Characteristics of 500,000 lb Spacecraft to Deliver a Jupiter Orbiter

a_o	I_c^*	T_C	t_{total}	W_{pl}	W_n	P	ℓ
units of g	sec	°K	years	lb	lb	Mw	ft
10^{-4}	3500	18,000	3.8	39,000	5200	27	3.7
3×10^{-4}	3010	15,400	3.1	8200	7700	46	4.2
10^{-4}	14,000	—	4.8	142,000	Nuclear Electric, 16.6 Kwe		
1.7×10^{-4}	8000	—	3.6	40,000			

a_o	N_{FC}	p	ϕ_c	M_F	α_*
units of g	nuclei/cm^3	atm	neut/cm^2 sec	kg	lg/kw
10^{-4}	12×10^{18}	4000	1.04×10^{14}	3.3	8
3×10^{-4}	4.3×10^{18}	1400	3.2×10^{14}	2.2	4.5

*Optimum to maximize payload fraction

Data: $\beta = 10^{-4}$, $f = 0.6$, $\zeta = 0.1$, $s = 0.05$, $x = 0.3$, $\theta = 0.3$, $\eta = 16$, $p_s = 30$ w/cm^3, $\phi_c/\phi_s = 1$, BeO Moderator, $\sigma_a(FC) = 1000$ barns

Note: zero payload with Nova or Rover

VI. DISCUSSION

Several general observations can be made about the prospects for advanced nuclear systems on the basis of the results obtained for the representative cases considered.

1. In the high-thrust application, there is a maximum possible specific impulse ratio of $\zeta^{-1/2}$, if regenerative cooling is the only mechanism for removing low quality heat from the engine.

In the case of hydrogen propellant this corresponds to a specific impulse of about 2200 sec, using 700 sec as a reference for the solid portion of the reactor.

2. At values of the thermal radiation parameter in the order of 10^{-2}, the gaseous systems offer large payload capability in single stage ground takeoff vehicles.

At $\beta = 10^{-2}$, the gaseous reactors can still produce specific impulses in the order of 1200 sec, and at these values the resulting systems compete very well with any other method of rocket propulsion.

3. Specific impulse ratios greater than $\zeta^{-1/2}$ can be achieved by the addition of a radiator, and application to interplanetary (low-acceleration) missions become feasible.

4. Low-acceleration missions to the near planets can be performed at flight times substantially less than that possible with electric systems. The near planets represent sufficiently easy missions (small velocity increments) that the very large specific impulse capability of the nuclear-electric systems allow them no great advantage over the gaseous systems with their relatively lower specific impulse. Consequently, the higher thrust-to-weight ratios of the latter yield marked reductions in

flight time. Even at values of $\beta \sim 10^{-2}$, initial accelerations of 10^{-2} g can be achieved, and these reduce trip times to coast periods. In a practical situation, however, such trips would most likely incorporate a nearly continuous propulsion period during both the planetary and heliocentric portion of the trajectory, and further reductions in flight time would become possible.

5. Low-acceleration missions to the far planets do not appear competitive with nuclear-electric systems unless the thermal radiation parameter can be reduced to 10^{-4} or less.

These higher performance missions place a premium on specific impulse, and unless β can be reduced to 10^{-4} or less, the radiator weight penalty does not permit the gaseous systems to operate at specific impulses anywhere near the values possible with nuclear-electric engines. Since the electric systems can attain specific impulses in the order of 10,000 sec, gaseous engines are apparently severely handicapped. The prospect of increasing the specific impulse to values of 5000 sec and up rests by and large on the thermal radiation characteristics of the gaseous mixture of fissionable species and propellant. In the case of the vortex containment method, typical tube dimensions, radial mass through-flows, and vortex characteristics (Ref. 1) require that the emissivity of the gas be in the order of 10^{-4} so as to yield β values as low as 10^{-4}.

Values of $\epsilon_c \sim 10^{-4}$ are not necessarily unreasonable when considered in terms of the present formulation of the thermal radiation problem. In this treatment (Ref. 4) we have used the form $\epsilon_c T_c^4$ to represent the thermal radiation from the cavity. Therefore, if the gas is to radiate at the value T_c, then to be consistent ϵ_c must indeed be very low. To complete the argument, it may be noted that in the event that

$\epsilon_c \sim 1$, the gas mixture would radiate like a black body, be very opaque, and the effective temperature of emission would be that of a layer approximately one optical depth inside the surface. At this depth the gas mixture would be some fraction of the central core temperature T_c.

These general remarks serve to point out that the prospect for achieving very high specific impulses with the aid of gas-phase fission heating is dependent upon the resolution of the thermal radiation problem, as well as upon the question of fuel containment. Further detailed analysis and especially experiments on the optical properties of suitable gaseous mixtures are required before the ultimate performance potential of these systems can be established.

The applicability of the gaseous system to the high-acceleration mission, however, appears to be apparent. The thermal radiation problem is less important in this instance since even small gains in specific impulse result in better vehicle performance than possible with staged-chemical or heat-exchanger type nuclear systems.

Thus if the containment problem can be solved, the high thrust application becomes an interesting possibility.

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